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**WIND TUNNEL INVESTIGATION TO DETERMINE PRESSURE
DISTRIBUTION CHARACTERISTICS OF A 0.03-SCALE
MODEL OF THE TITAN III/MOL LAUNCH CONFIGURATION DURING
THE ABORT SEQUENCE AT MACH NUMBERS FROM 0.60 TO 3.00**

D. A. MacLanahan, Jr. and M. L. Homan
ARO, Inc.

July 1967

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AF 40(600)1200

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FOREWORD

The work reported herein was done at the request of SAMSO (SSBD), Air Force Systems Command (AFSC), for the Martin Company, under Program Element 63409404, System 632A and Program Element 63409304, System 623A.

The results of the test were obtained by ARO, Inc. (a subsidiary of Sverdrup & Parcel and Associates, Inc.), contract operator of the Arnold Engineering Development Center (AEDC), AFSC, Arnold Air Force Station, Tennessee, under Contract AF 40(600)-1200. The test was conducted from March 13 to April 24, 1967, under ARO Project No. PT0732, and the manuscript was submitted for publication on June 16, 1967.

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This technical report has been reviewed and is approved.

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Directorate of Test

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Colonel, USAF
Director of Test

ABSTRACT

A 0.03-scale model of the Titan III/Manned Orbiting Laboratory (MOL) launch vehicle was tested in Tunnels 16T and 16S of the Propulsion Wind Tunnel Facility at Mach numbers from 0.6 to 3.0 to determine pressure distributions on the airborne vehicle during the abort sequence. Test results show that the jet, simulating thrust termination, forces the solid-propellant motor induced normal shock forward. Disturbances caused by the jet produce an increase in the pressure coefficient in the region ahead of the jet and a decrease in the pressure coefficient in the region aft of the jet.

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NOMENCLATURE

C_p	Pressure coefficient, $\frac{p_L - p_\infty}{q_\infty}$
D	Core diameter (reference diameter), 0.30 ft
M_∞	Free-stream Mach number
p_c	Chamber pressure for the thrust termination simulation system, psfa
p_L	Surface pressure measured at the pressure orifice, psfa
p_∞	Free-stream static pressure, psfa
q_∞	Free-stream dynamic pressure, psf
x	Distance of pressure orifice location from model nose, ft
α	Model angle of attack, deg
β	Model angle of sideslip, deg
θ	Radial orientation of surface pressure orifice (see Fig. 7), deg

SECTION I INTRODUCTION

The Titan III/MOL (Manned Orbiting Laboratory) launch system combines a Titan 120-in. -diam core with two 120-in. -diam, seven-segment, strap-on solid-propellant motors (SRM) with MOL. The MOL is a 120-in. -diam cylinder with a Gemini B attached to the forward bulkhead. The flight crew occupies the Gemini B during launch.

During the time from lift-off, until SRM burn out and are jettisoned, a condition might occur caused by some malfunction of the launch vehicle that would make it necessary to initiate an abort of the mission. An abort sequence is initiated by thrust termination, opening exhaust ports in the nose of SRM. The time available to the flight crew to escape from the vehicle is, in part, dependent upon local airloads which determine the structural integrity.

A 0.03-scale model of the Titan III/MOL launch vehicle was tested in the Propulsion Wind Tunnel (PWT) Transonic (16T) and Supersonic (16S) circuits to obtain surface pressure data on the vehicle during a simulated abort sequence. High pressure air was used to simulate the rocket exhaust from the thrust termination ports, which were located on the nose of SRM. No attempt was made to simulate the normal rocket exhaust. Data were obtained at Mach numbers from 0.6 to 3.0 for model angles of attack and angles of sideslip from -35 to 35 deg.

This report is concerned with the pressure phase of this test. Only the significant test results are presented. The complete test data were forwarded to the user and are available at AEDC. The force phase of this test is presented in Ref. 1.

SECTION II APPARATUS

2.1 TEST FACILITIES

Tunnel 16T is a closed-circuit, continuous flow wind tunnel capable of operating at Mach numbers from 0.55 to 1.60. The tunnel is capable of operating over a stagnation pressure range from approximately 160 to 4000 psfa and over a stagnation temperature range from 80 to 160°F. The tunnel specific humidity is controlled by removing the tunnel air and supplying conditioned makeup air from an atmospheric dryer. Perforated walls in the test section allow continuous operation through the Mach number range with a minimum of wall interference.

Tunnel 16S is a closed-circuit, continuous flow wind tunnel capable of operating at Mach numbers from 1.65 to 3.20. The tunnel is capable of operating over a stagnation pressure range from 100 to approximately 1800 psfa. The test section stagnation temperature can be controlled through the range from 100 to 650°F. The tunnel specific humidity is controlled by removing tunnel air and supplying conditioned makeup air from an atmospheric dryer.

Details of the test sections showing model location and support strut arrangement are presented in Figs. 1 through 4. A more extensive description of each tunnel and its operating characteristics is contained in Ref. 2.

2.2 MODEL GEOMETRY

The Titan III/MOL launch vehicle combines a Titan III 120-in. -diam core and two 120-in. -diam, seven-segment, strap-on SRM with MOL. The MOL consists of a 120-in. -diam cylinder mounted aft of a Gemini B spacecraft. A 0.03-scale pressure model of the basic launch vehicle was tested in each tunnel. Model details are presented in Figs. 5 and 6.

To obtain the required angle-of-attack and sideslip range for these tests, the test article was supported in the wind tunnel by means of a remotely operated, cantilevered sting attached to the basic sting support system. This auxiliary system is capable of operation within the range from +30 to -11 deg. Wind tunnel installations are shown in Figs. 1 through 4.

High pressure air was used to simulate the solid-propellant jets that would be exhausting through the thrust termination ports on the nose of SRM during the abort sequence. Each SRM contains an internal tank which provides a stilling chamber for the air supplied to the thrust termination ports (Fig. 6). The air is supplied through the aft end of each SRM from a high pressure supply located outside the tunnel shell.

2.3 INSTRUMENTATION

The model was instrumented with 233 surface pressure orifices for measurement of steady-state pressures. The chamber pressure of the thrust termination simulation system was measured by two model-mounted, 500-psia transducers. All pressure orifice locations are shown in Fig. 7. All output signals from pressure transducers, angle of attack, and roll systems were digitized and code punched on paper tape for reduction by a Raytheon 520 computer.

SECTION III TEST DESCRIPTION

3.1 PROCEDURE

Angles of attack (α) and angles of sideslip (β) were obtained by remotely pitching and rolling the main support mechanism and pitching the auxiliary mechanism. The model and sting were manually rolled, at the joint where the main support mechanism meets the auxiliary sting, to any one of the 90-deg roll positions that will give the required angle-of-attack and angle-of-sideslip combinations for that position. The range of angle of attack and angle of sideslip thus obtainable was from -35 to +35 deg. Since the model was instrumented to only one quadrant, 0 to 90 deg, the protuberances on MOL were relocated to obtain data in one additional quadrant, 180 to 270 deg.

Data were obtained while holding Mach number and thrust termination chamber pressure ratio (p_c/p_∞) constant and varying α and β within the available range at each model roll position. The model was tested at Mach numbers from 0.6 to 1.4 in Tunnel 16T and from 1.8 to 3.0 in Tunnel 16S. Reynolds number per foot varied from 1.9×10^6 to 4.0×10^6 in Tunnel 16T and from 1.0×10^6 to 2.0×10^6 in Tunnel 16S.

3.2 PRECISION OF MEASUREMENT

The estimated precision of measurement is as follows:

Angle of attack or sideslip (neglecting airload deflection)	± 0.20 deg
Mach number 0.60 to 1.10	± 0.003
Mach number 1.20 to 1.40	± 0.010
Mach number 1.80 to 3.10	± 0.020
Tunnel static pressure	± 1 psf

The maximum uncertainty for pressure coefficient was ± 0.02 .

SECTION IV RESULTS AND DISCUSSION

The purpose of this phase of this test was to obtain surface pressure data along the model centerbody of a 0.03-scale model of the basic launch

configuration of a Titan III/MOL during a simulated abort sequence at Mach numbers from 0.6 to 3.0. Since the model was instrumented in only one quadrant, 0 to 90 deg, the protuberances were relocated by a 180-deg circumferential displacement in order to obtain data in the 180- to 270-deg quadrant. With the exception of one small protuberance (see Fig. 5), the core configuration is symmetrical about the yaw plane. For model attitudes of equal and opposite sign, it was possible to use the fixed pressure instrumentation of the one quadrant to obtain data for all quadrants. For all test Mach numbers, the thrust termination simulation jet pressure ratio (p_c/p_∞) was chosen to correspond to the conditions of a nominal trajectory. The pressure ratio for the nominal trajectory is presented in Fig. 8.

4.1 EFFECTS OF JET

Centerline pressure coefficient distributions, for $\alpha = 0$, $\beta = 0$ deg, with and without thrust termination simulation (jet-on and jet-off), are presented in Fig. 9. At each free-stream Mach number, jet-on operation resulted in an increase in pressure coefficient upstream of the jet exit and a decrease in pressure coefficient downstream of the jet exit. The jet-on pressure distributions are similar to those observed with a jet issuing normal to a flat plate (see Ref. 3). At subsonic Mach numbers, the pressure differences, jet on to jet off, immediately downstream of the jet exit are much greater than those upstream of the jet exit, but at supersonic Mach numbers, the opposite trend predominates.

The interference of the thrust termination jet extends over the complete model at all Mach numbers, except for a region near the nose of the centerbody at supersonic Mach numbers. The extent of this latter region is a function of Mach number, and the interference extends least forward for Mach number 1.4. The forward progression of the interference increases as jet pressure ratio (p_c/p_∞) increases (increased Mach number) and is attributable to the forward movement of the SRM induced normal shock caused by increased jet plume expansion. Schlieren photographs at Mach number 1.2 (Fig. 10) show the movement of the SRM induced normal shock as a result of the jet. Figure 10 also shows the turbulence caused by the jet.

4.2 EFFECTS OF MODEL ATTITUDE

Centerline pressure coefficients for various angles of attack and angles of sideslip are shown in Fig. 11 for nominal trajectory pressure ratios. Top and bottom centerline surface pressures are presented for

angle of sideslip of 0 deg and angle of attack of 30 deg. Right and left centerline pressures are presented for angle of attack of 0 deg and angle of sideslip of 30 deg. The dashed lines represent 0-deg angle-of-attack and sideslip data for comparison (symbols omitted for clarity). In the region upstream of the jet exit, the pressure distributions at $\alpha = 30$ and $\beta = 0$ deg are similar to those for $\alpha = 0$ and $\beta = 30$ deg. Downstream of the jet exit, an angle of attack of 30 deg produced much greater pressure differences across the model than did an equal angle of sideslip.

Circumferential centerbody pressure distribution at two model stations ($x/D = 4.182$ and 6.474) for various angles of attack, angles of sideslip, and Mach number are shown in Fig. 12. These pressure distributions indicate that the longitudinal pressure distributions previously presented at 0-, 90-, 180-, and 270-deg circumferential locations are representative of the maximum and minimum pressures which occur on the centerbody.

4.3 EFFECTS OF PRESSURE RATIO

The effect of jet pressure ratio on longitudinal pressure distribution at zero angle of attack and sideslip is shown in Fig. 13 for two off-trajectory pressure ratios at Mach numbers 1.4 and 2.2. Increasing the pressure ratio resulted in an increase in the pressure coefficient upstream of the jet exit and a decrease in the pressure coefficient downstream of the jet exit. Increasing the pressure ratio resulted in a forward movement of the SRM induced normal shock.

Variations of pressure coefficient distributions with jet pressure ratio at two model attitudes are shown in Fig. 14. With the model at an angle of attack of 30 deg for the bottom (windward) centerline orifices, increasing jet pressure ratio resulted in large changes in pressure distribution. For the top (lee) centerline orifices, varying the pressure ratio had negligible effect on the pressure distribution. With the model at an angle of sideslip of 30 deg, for Mach number 2.2, varying jet pressure ratio resulted in small local pressure variations.

SECTION V CONCLUDING REMARKS

Centerbody surface pressure data were obtained for a Titan III/MOL during a simulated abort sequence at Mach numbers from 0.6 to 3.0. The results of the investigation are summarized below:

1. Thrust termination simulation (jet-on) produced an increase in pressure coefficient upstream of the jet exit and a decrease in pressure coefficient downstream of the jet exit. The SRM induced normal shock moves forward with jet on, thereby extending the region of interference over the model.
2. Increasing the jet pressure ratio (p_c/p_∞) resulted in an increase in pressure difference, jet on to jet off, in the region ahead of the ports and a decrease in pressure difference in the region behind the ports. The SRM induced normal shock is moved farther forward by increasing the jet static pressure ratio.
3. The SRM induced normal shock moves forward with Mach number, thereby increasing the extent of interference on the model. This is attributed to expansion of the exhaust plume produced by increasing exit pressure ratio.

REFERENCES

1. Homan, M. L. and MacLanahan, D. A., Jr. "Wind Tunnel Investigation to Determine Aerodynamic Characteristics of a 0.03-Scale Model of the Titan III/MOL Launch Configuration during the Abort Sequence at Mach Numbers from 0.60 to 3.00". AEDC-TR-67- , July 1967.
2. Test Facilities Handbook (6th Edition). "Propulsion Wind Tunnel Facility, Vol. 5." Arnold Engineering Development Center, November 1966.
3. Reichenau, David E. A. "Interference Effects of Cold and Hot Rocket Exhausts Issuing Normal to the Airstream from a Flat Plate at Free-Stream Mach Numbers from 0.6 to 1.4." AEDC-TR-66-127, June 1966.

**APPENDIX
ILLUSTRATIONS**

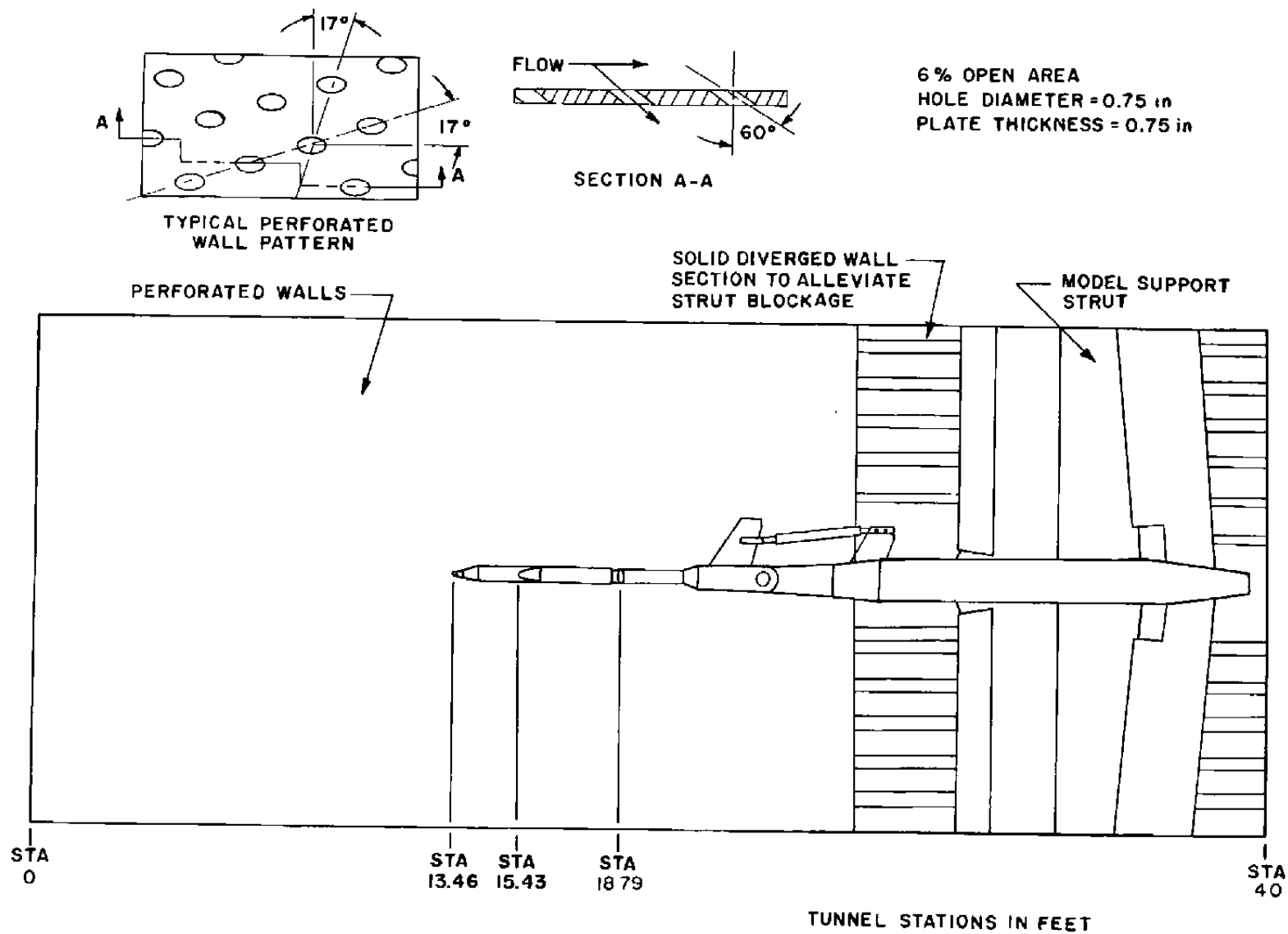


Fig. 1 Location of Model Installed in Tunnel 16T

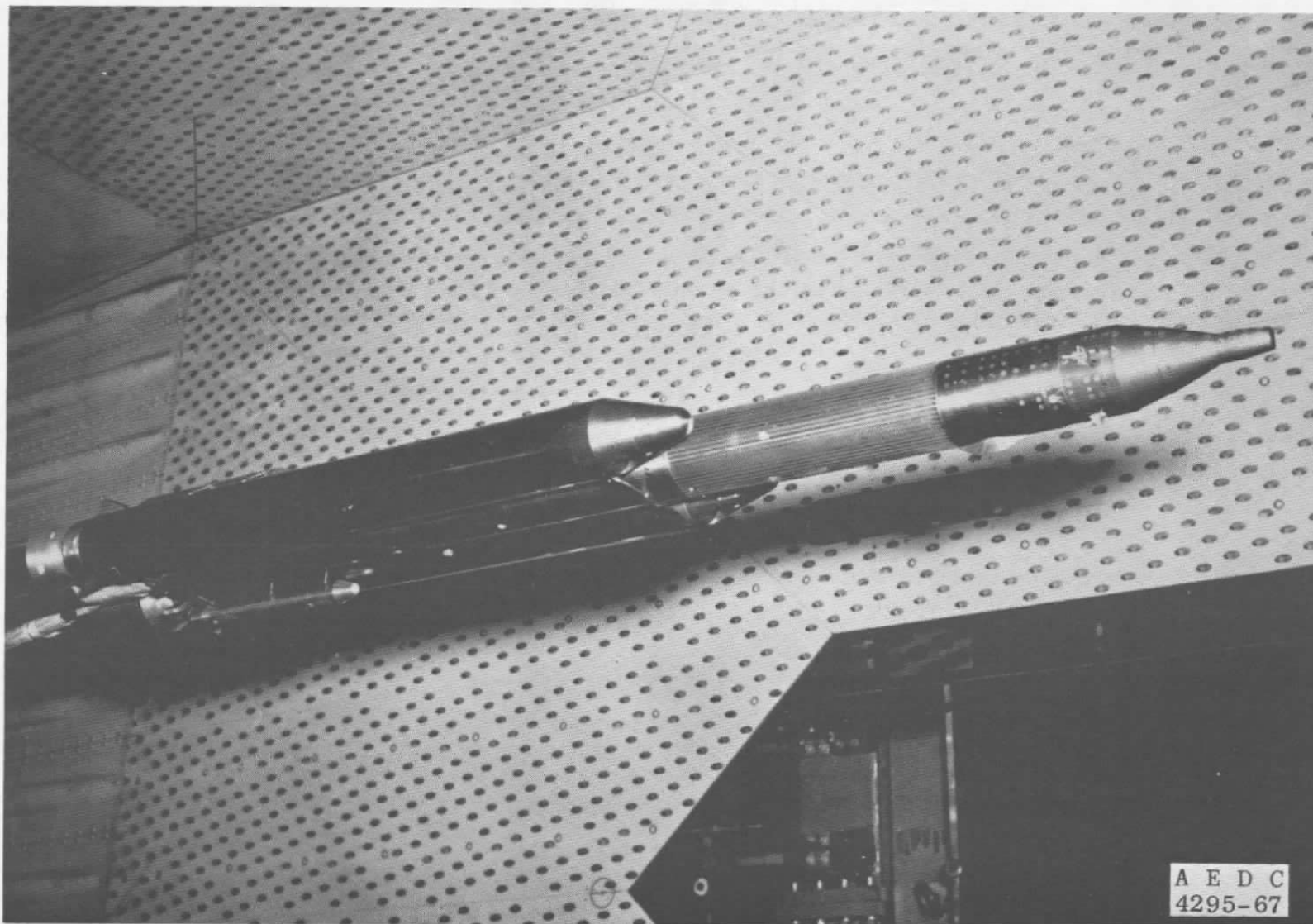


Fig. 2 Photograph of Model Installed in Tunnel 16T

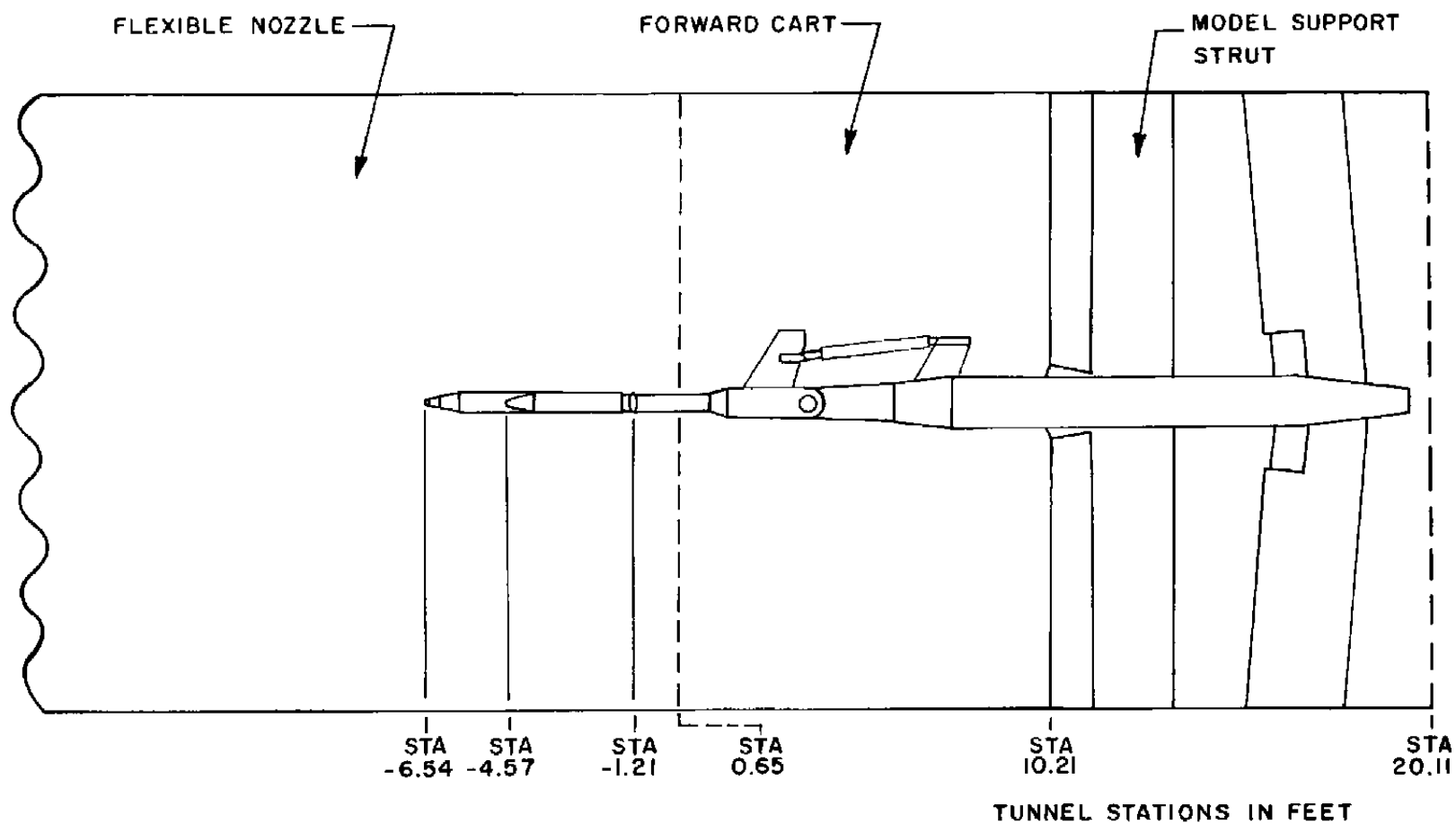


Fig. 3 Location of Model Installed in Tunnel 16S

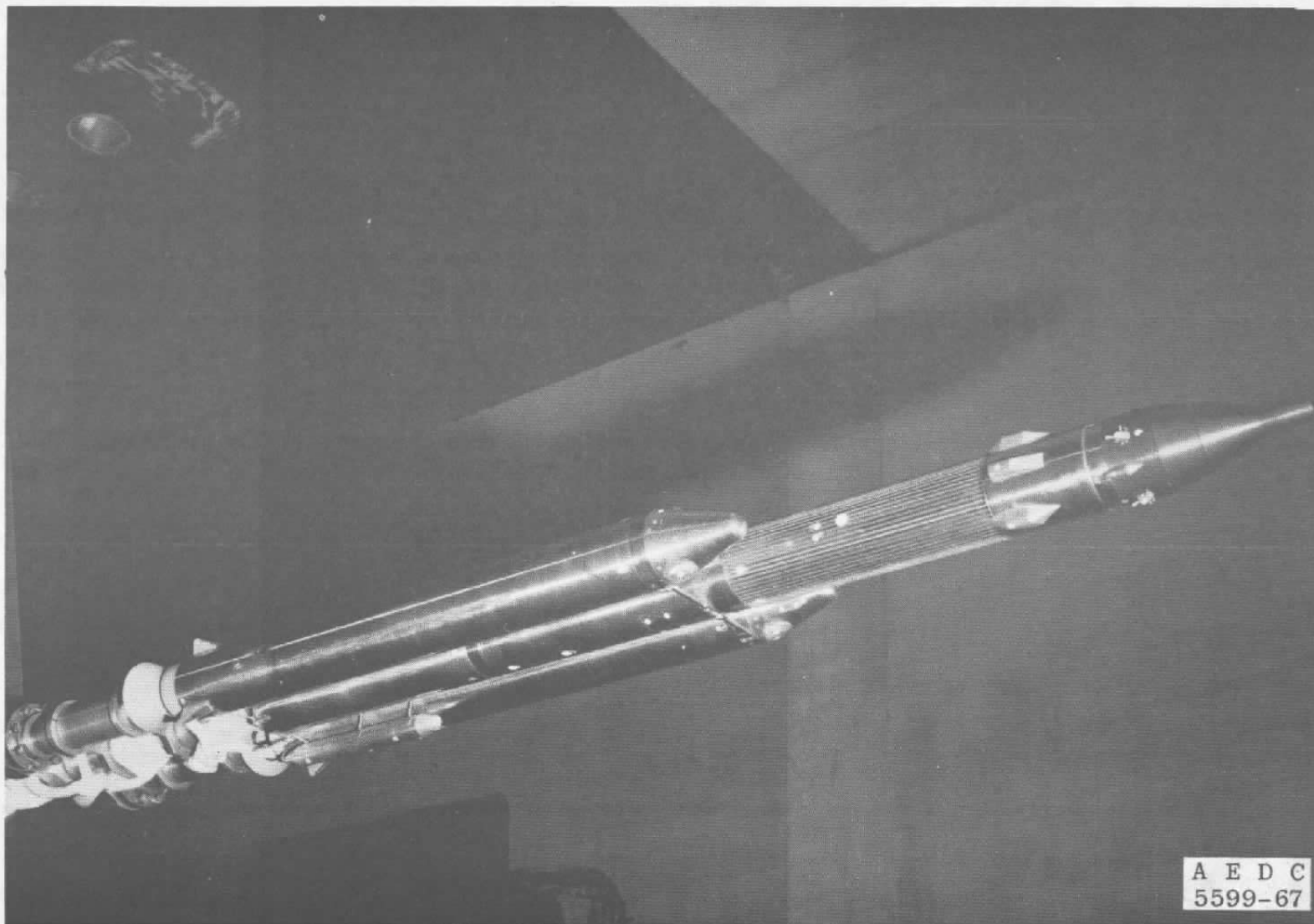
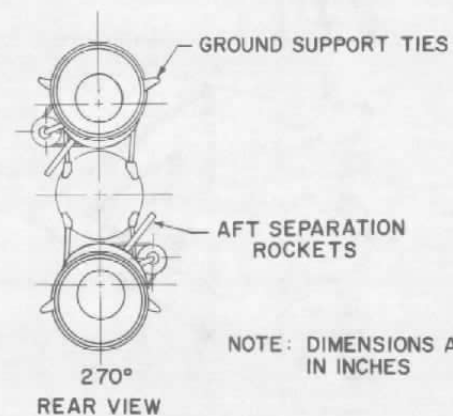
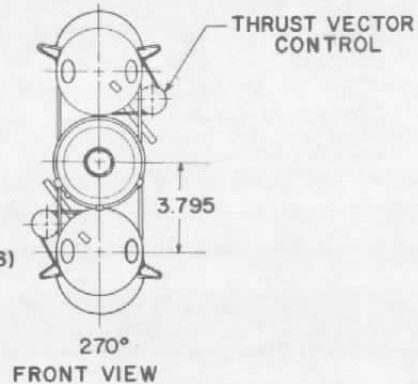
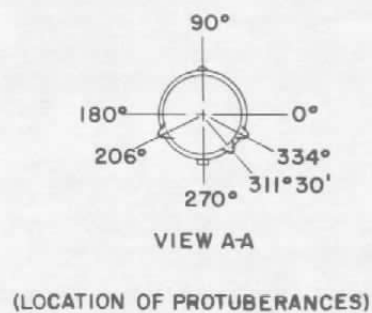
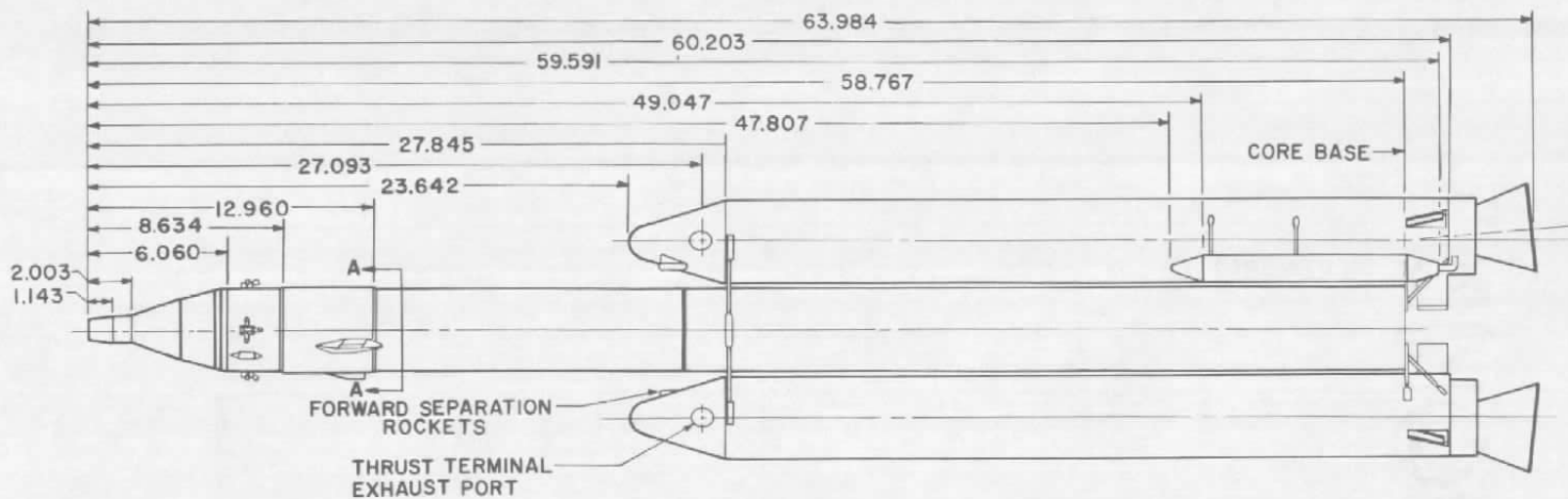


Fig. 4 Photograph of Model Installed in Tunnel 16S



NOTE: DIMENSIONS ARE IN INCHES

Fig. 5 Model Details

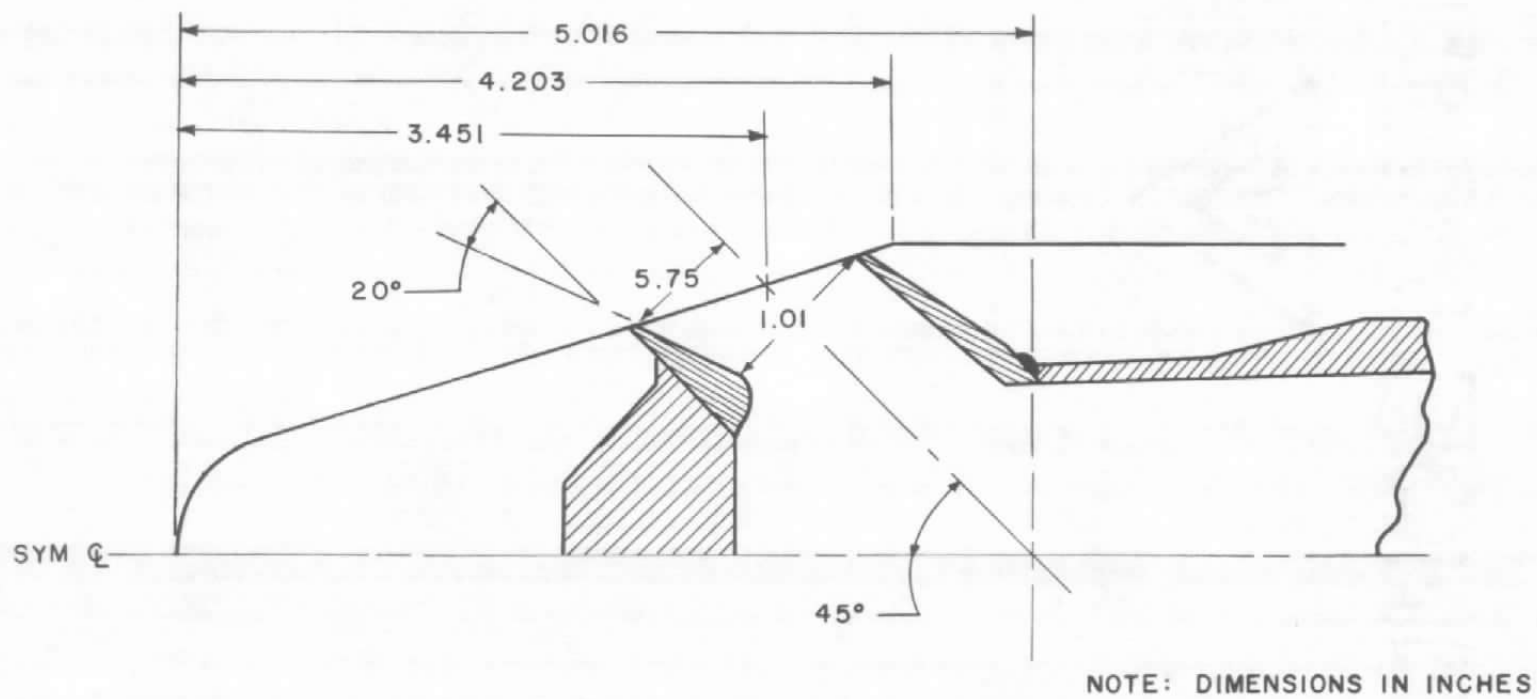
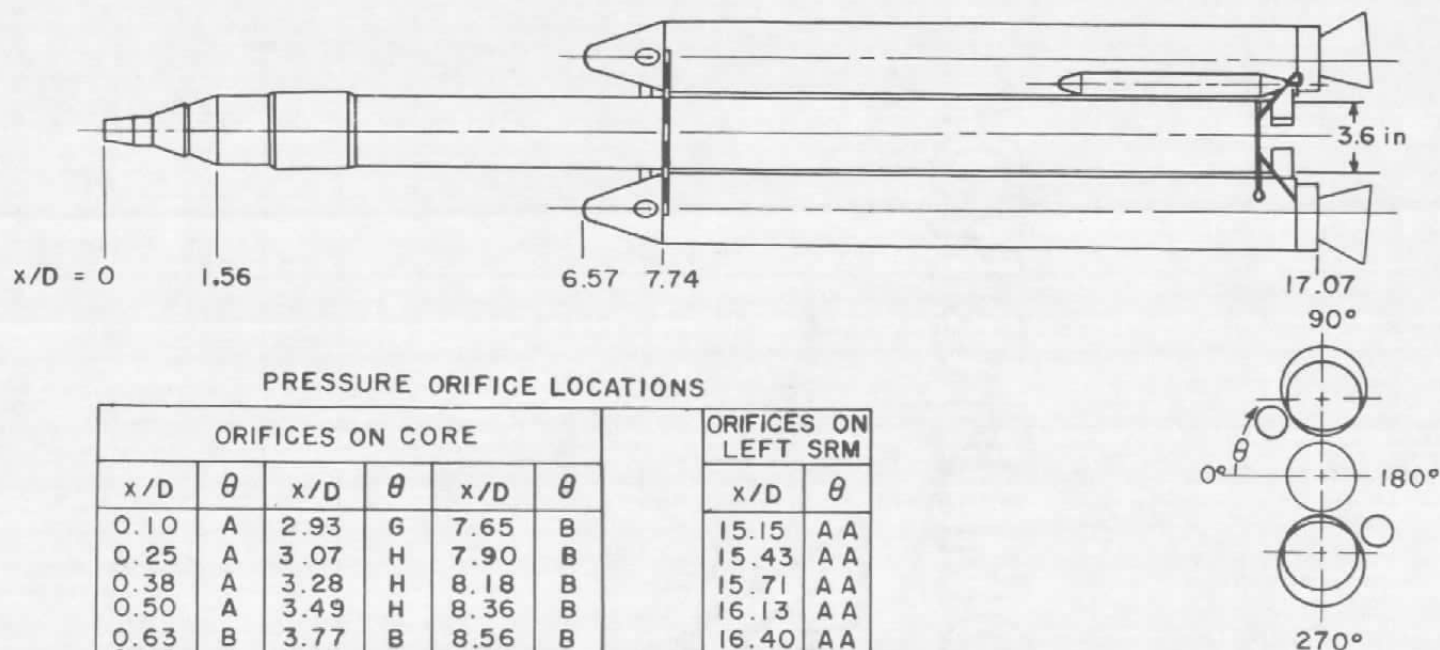


Fig. 6 Thrust Termination Port Details



PRESSURE ORIFICE LOCATIONS

ORIFICES ON CORE						ORIFICES ON LEFT SRM	
x/D	θ	x/D	θ	x/D	θ	x/D	θ
0.10	A	2.93	G	7.65	B	15.15	AA
0.25	A	3.07	H	7.90	B	15.43	AA
0.38	A	3.28	H	8.18	B	15.71	AA
0.50	A	3.49	H	8.36	B	16.13	AA
0.63	B	3.77	B	8.56	B	16.40	AA
0.85	B	4.18	B	9.35	B	16.82	AA
1.07	B	4.60	B	10.35	B		
1.21	B	5.02	B	11.27	B		
1.54	B	5.43	B	12.52	B		
1.64	B	5.85	B	13.35	B		
1.78	C	6.36	B	14.43	B		
1.89	C	6.47	B	14.74	J		
2.10	C	6.68	B	15.02	J		
2.32	D	6.89	B	15.29	J		
2.46	D	7.10	B	15.68	B		
2.65	E	7.35	B	15.85	J		
2.79	F	7.47	I	16.13	J		
		7.57	I				

PRESSURE LEGEND

A: 0, 45, 90 deg
 B: 0, 30, 60, 90 deg
 C: 0, 30, 60, 90, 120, 142 deg
 D: 0, 30, 60, 90, 120, 131.5, 142 deg
 E: 0, 30, 60, 90, 131.5 deg
 F: 0, 20, 40, 60, 76, 90 deg
 G: 0, 10, 20, 40, 60, 76, 90 deg
 H: 0, 10, 20, 30, 40, 60, 76, 90 deg
 I: 0, 30, 60, 80 deg
 J: 0, deg
 AA: 0, 270 deg

Fig. 7 Surface Pressure Orifice Locations

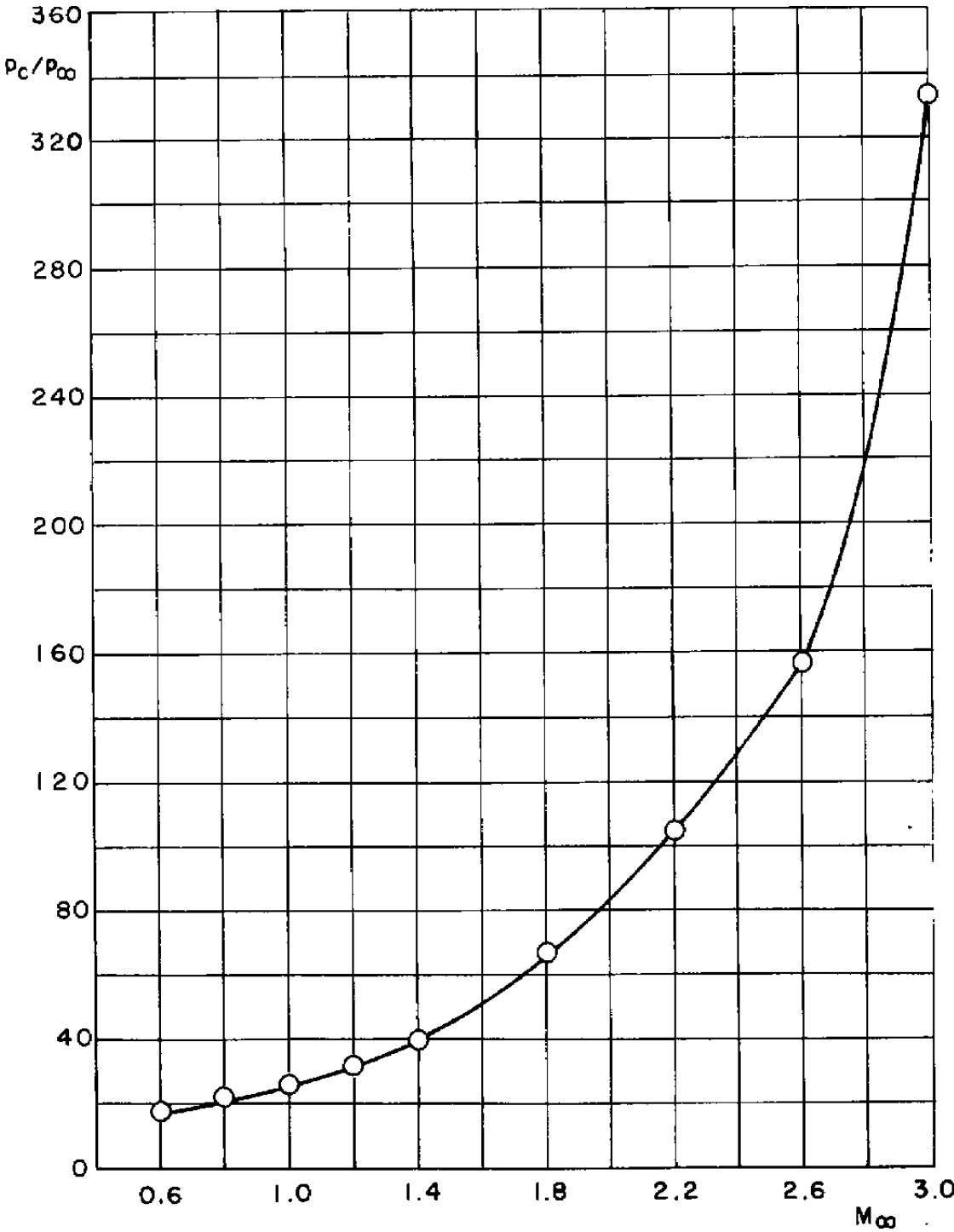


Fig. 8 Pressure Ratio Required for Nominal Trajectory

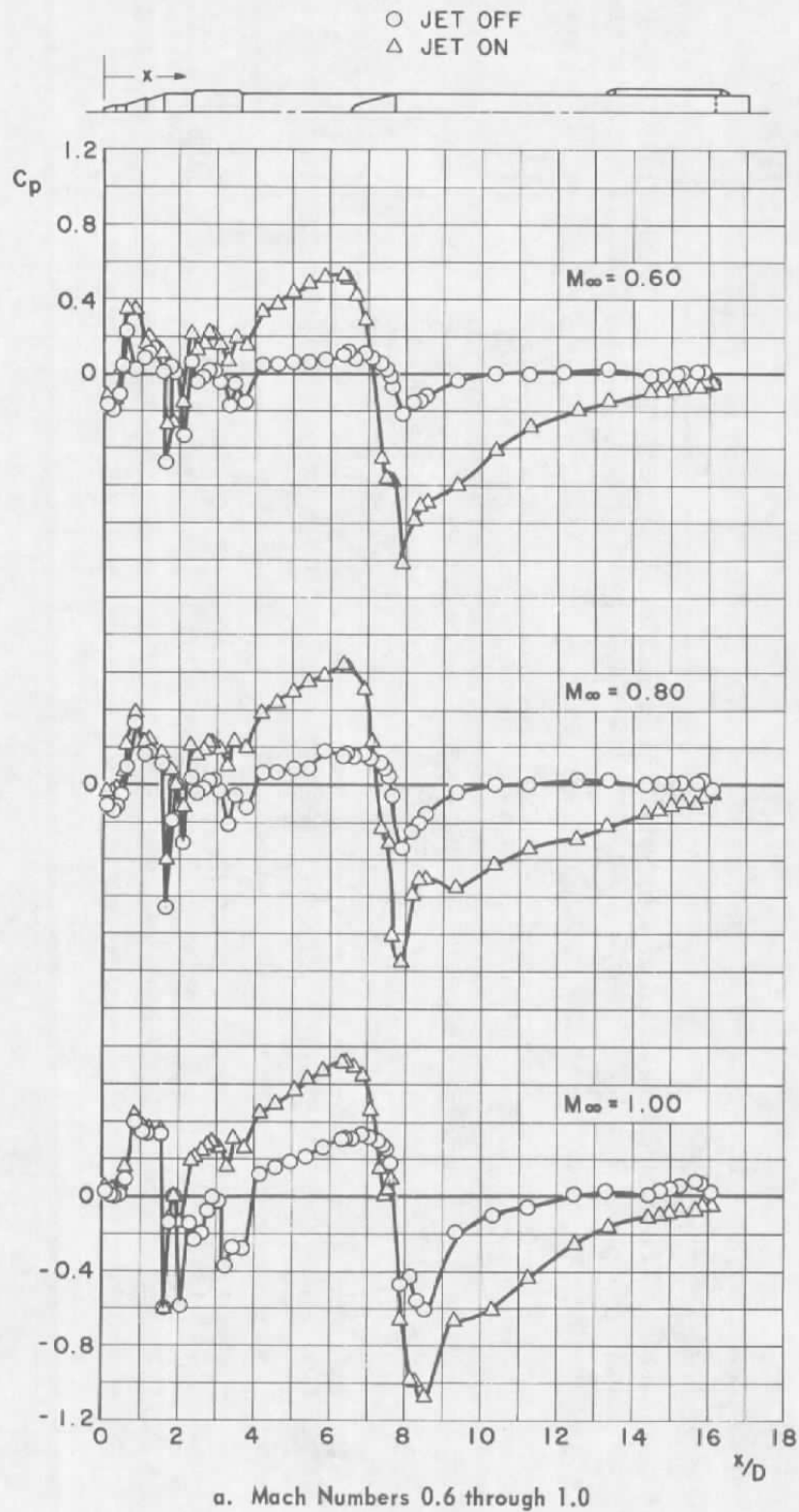
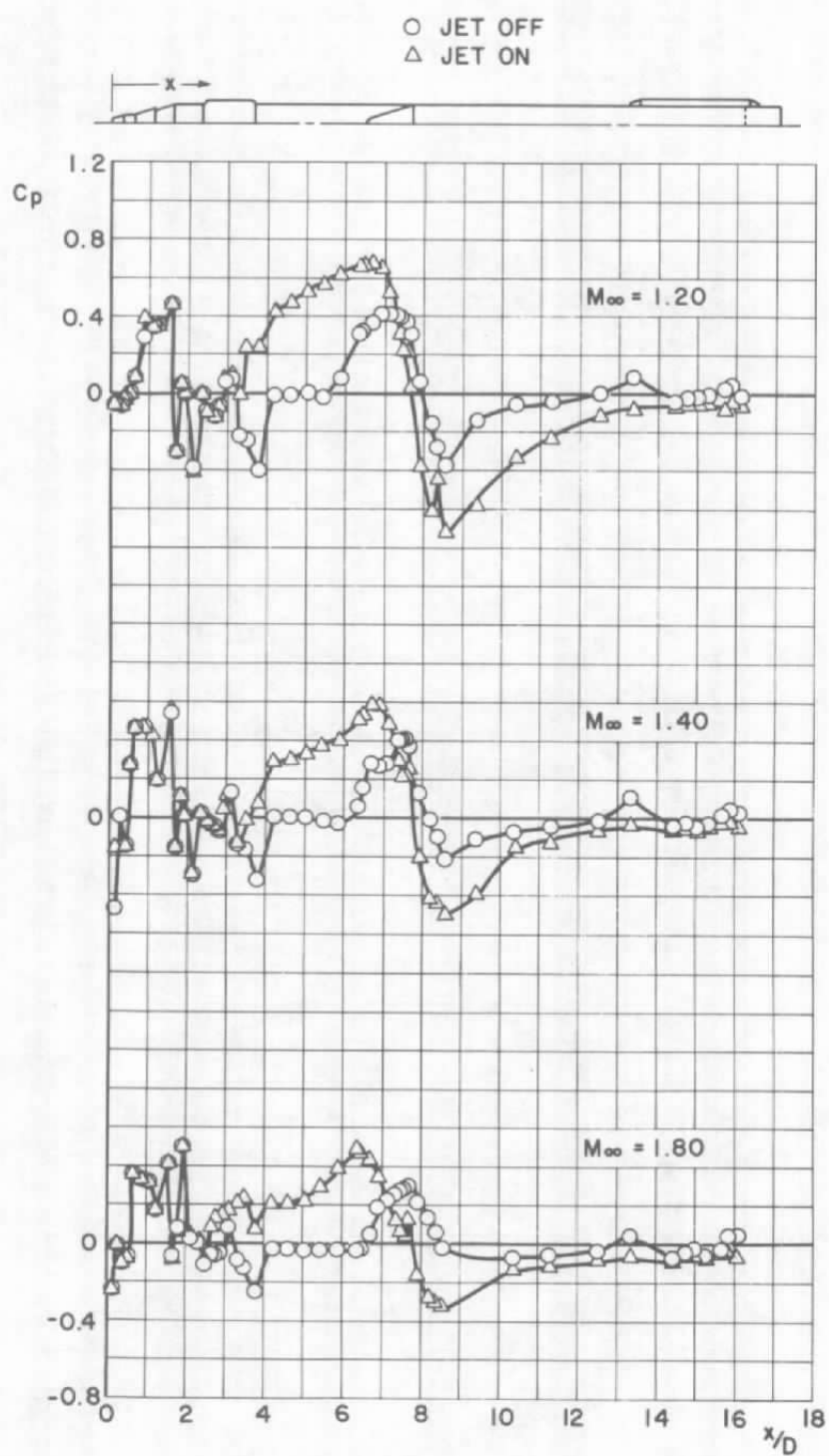
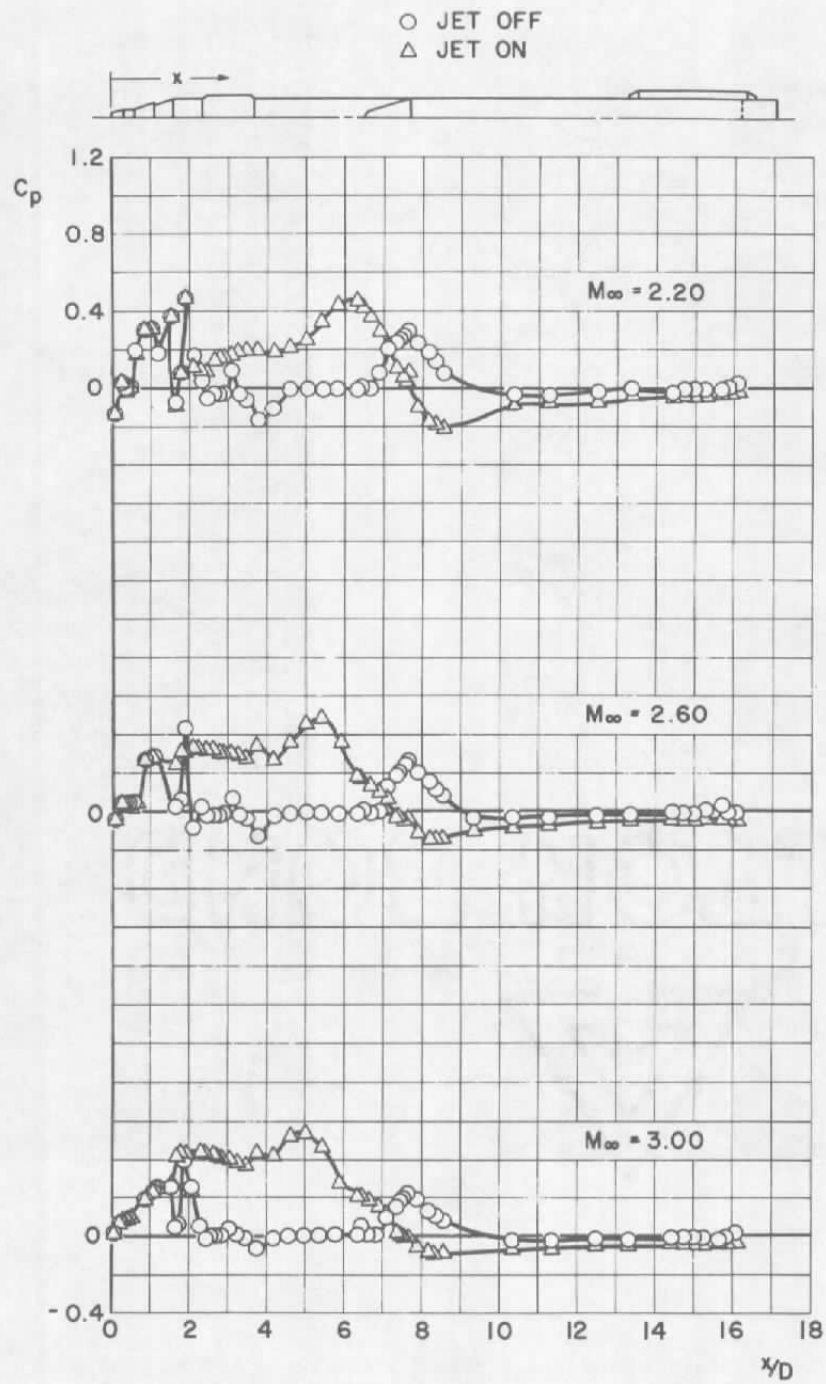


Fig. 9 Centerbody Pressure Distribution with Thrust Termination Simulation for Nominal Trajectory Pressure Ratio, $\theta = 180$ deg, $\alpha = 0$ deg, and $\beta = 0$ deg



b. Mach Numbers 1.2 through 1.8
Fig. 9 Continued



c. Mach Numbers 2.2 through 3.0

Fig. 9 Concluded

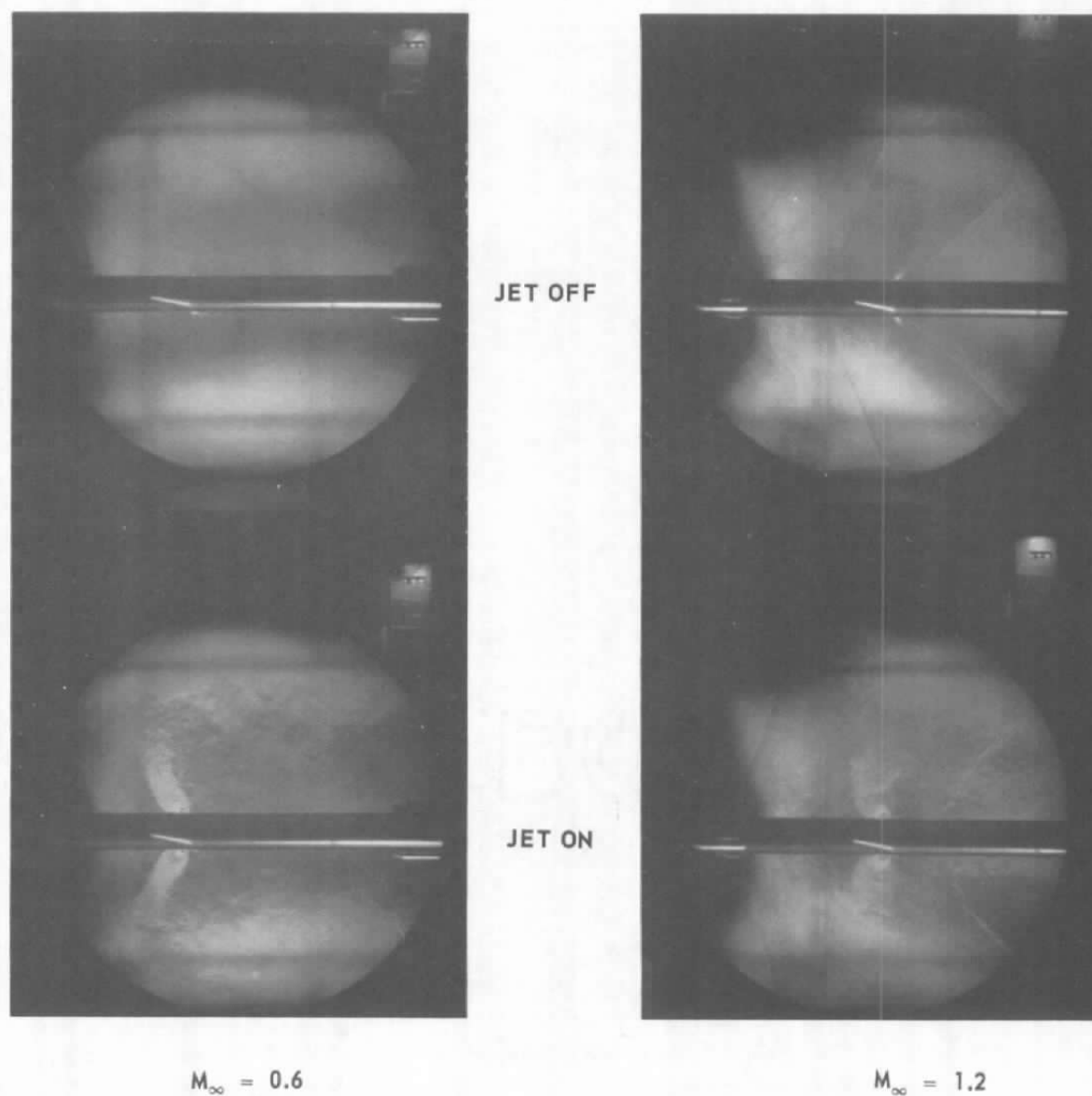
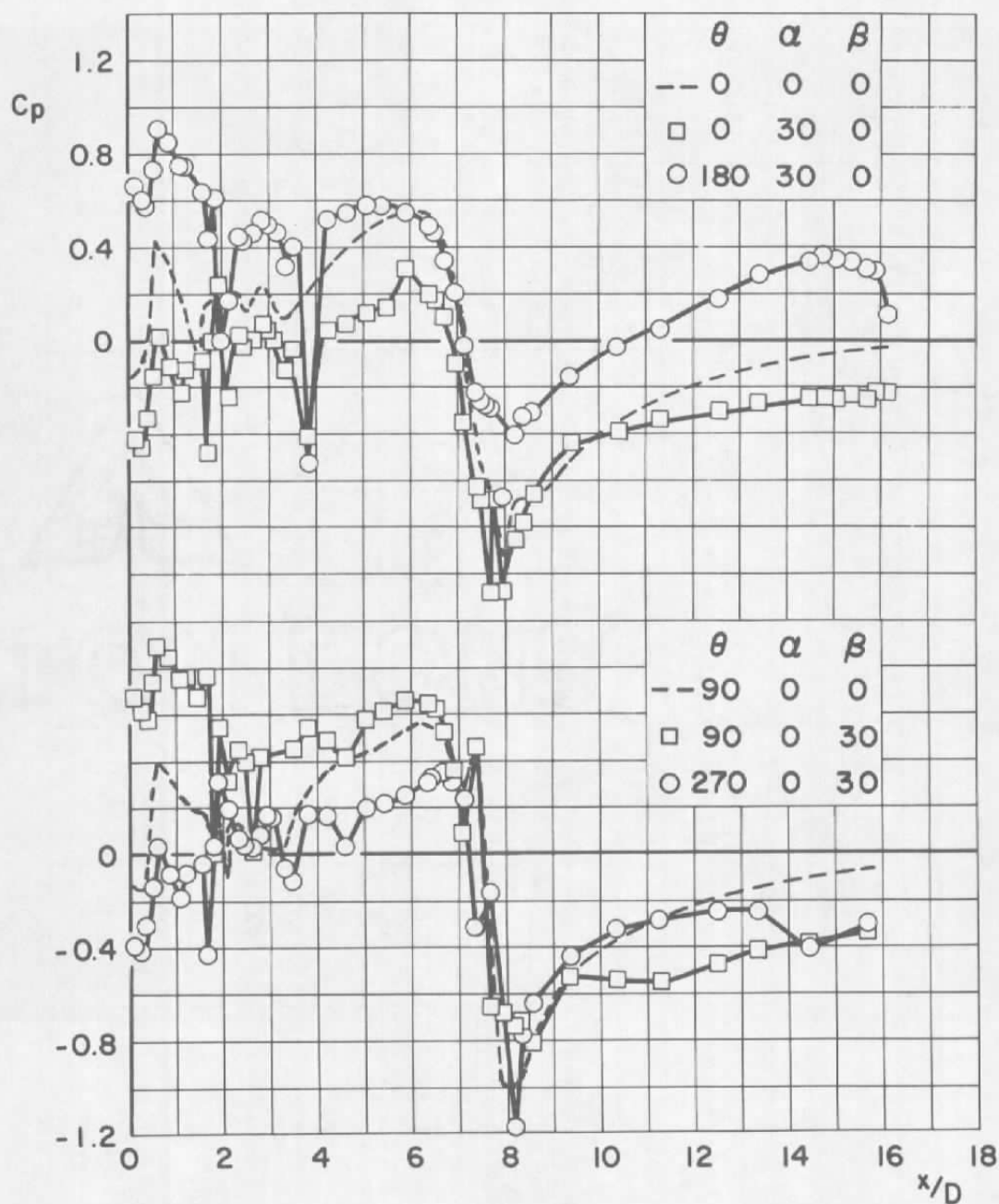
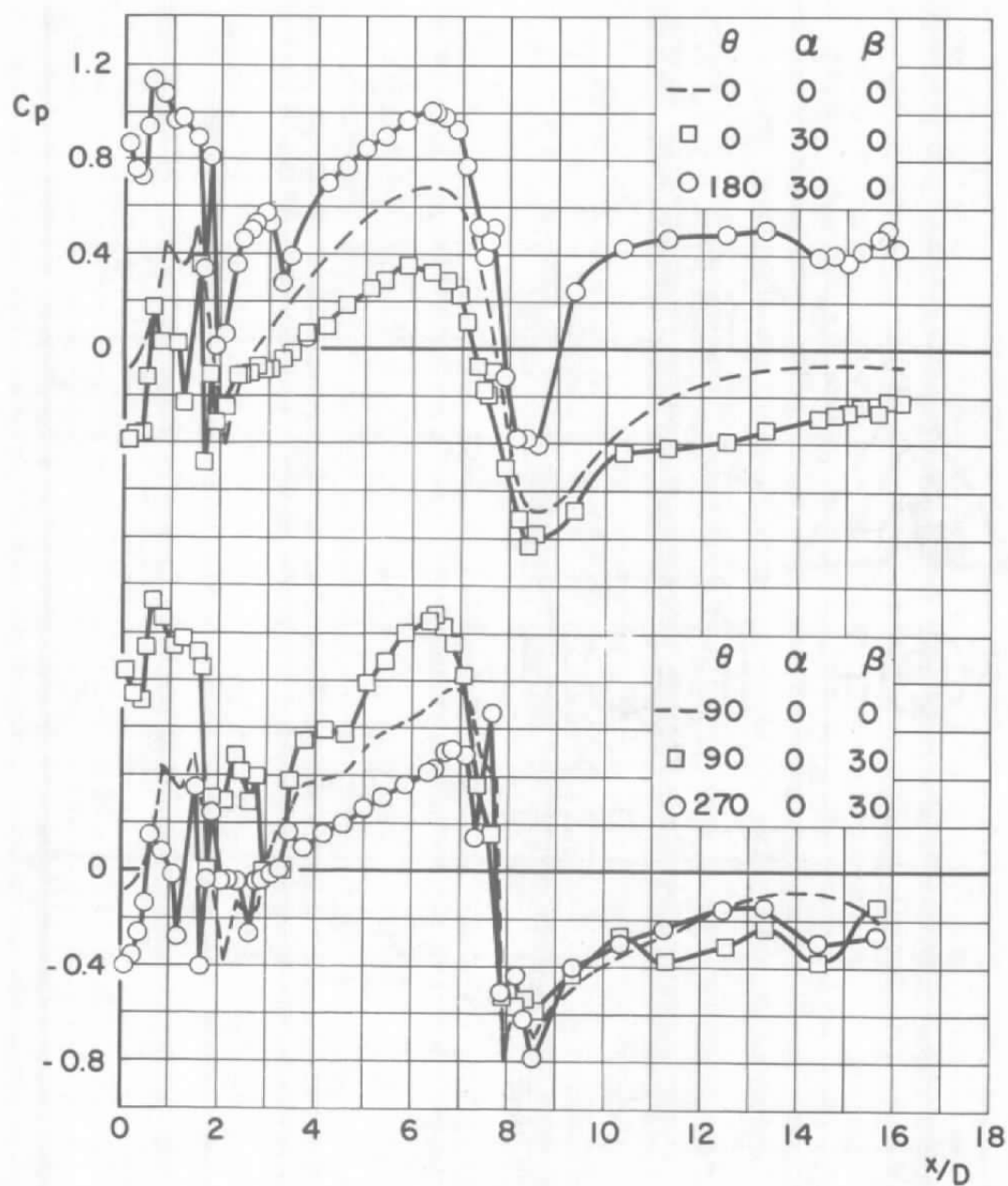


Fig. 10 Schlieren Photographs of the SRM Noses at $\alpha = 0$ deg and $\beta = 0$ deg with and without Thrust Termination Simulation at Mach Numbers 0.6 and 1.2

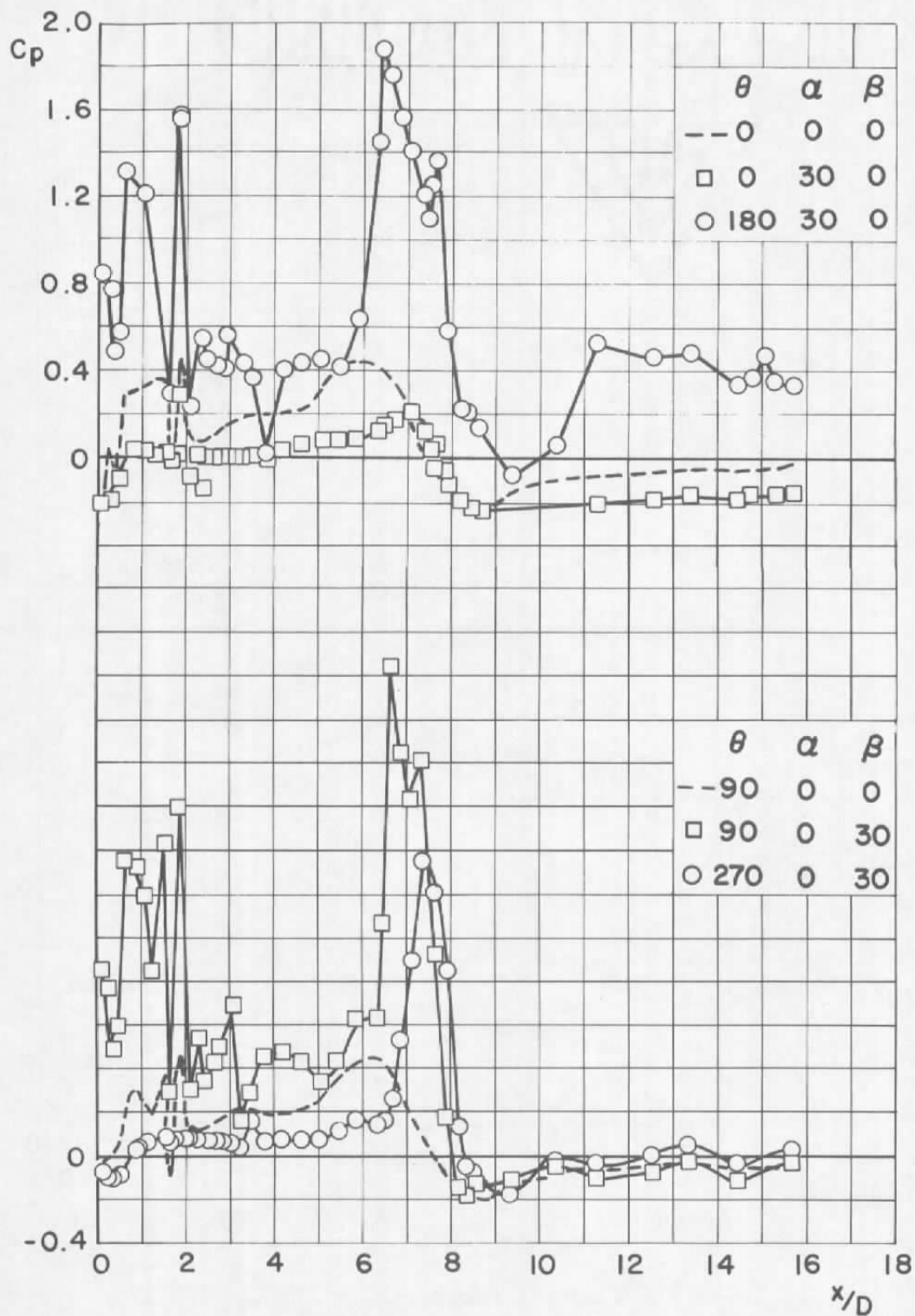


a. Mach Number 0.6

Fig. 11 Centerbody Pressure Distribution for Nominal Trajectory Pressure Ratio



b. Mach Number 1.2
Fig. 11 Continued



c. Mach Number 2.2

Fig. 11 Concluded

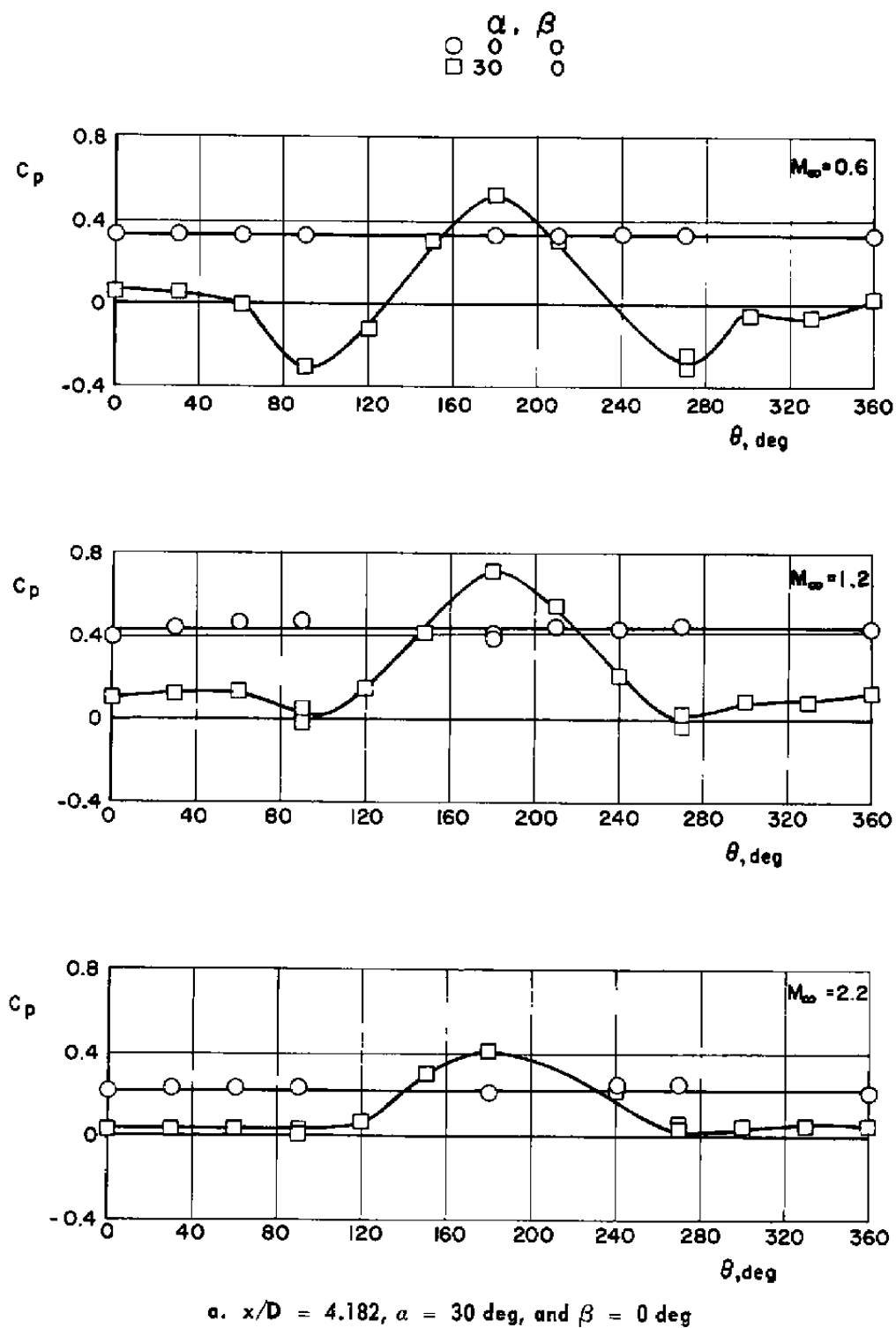
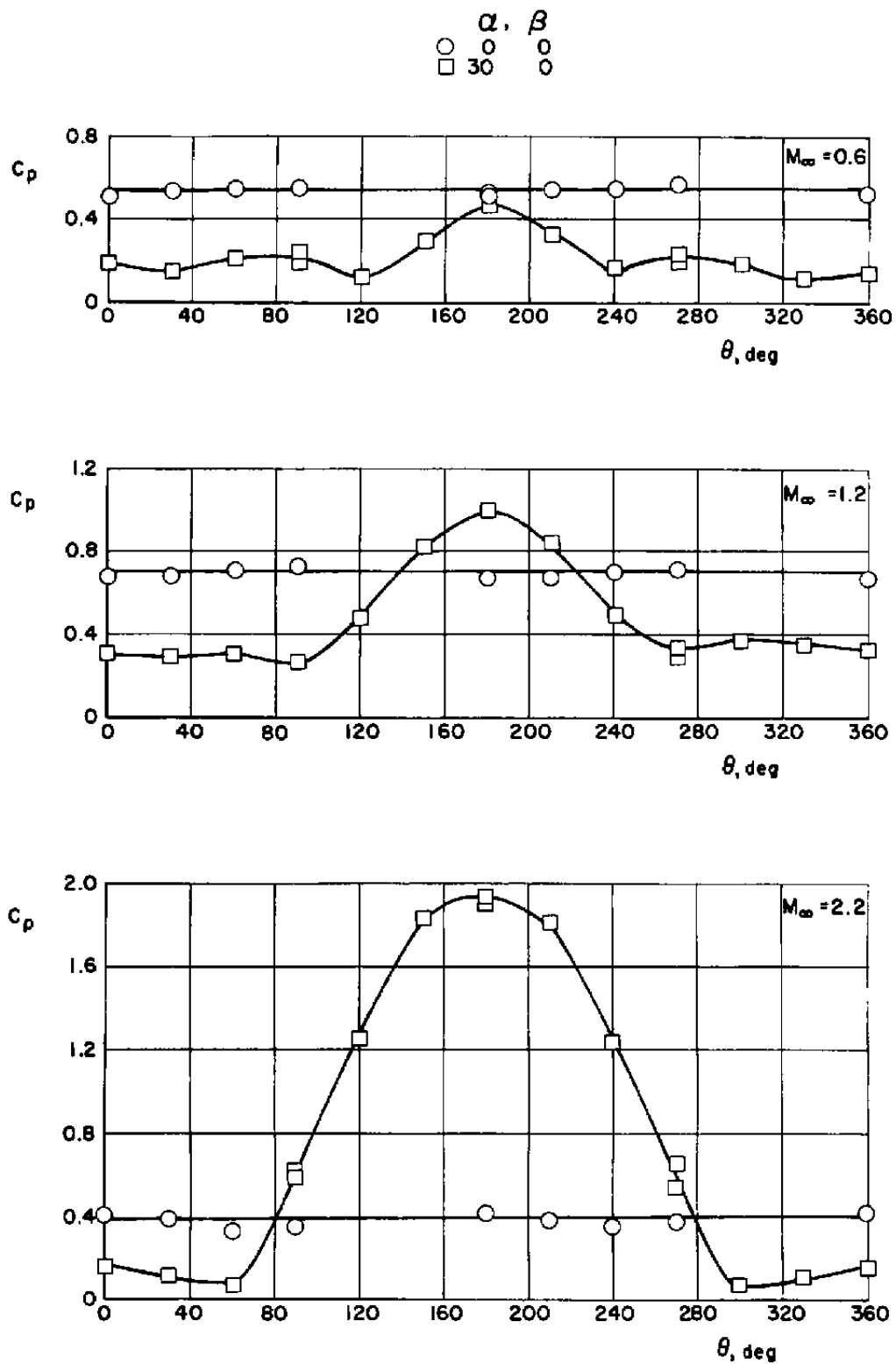


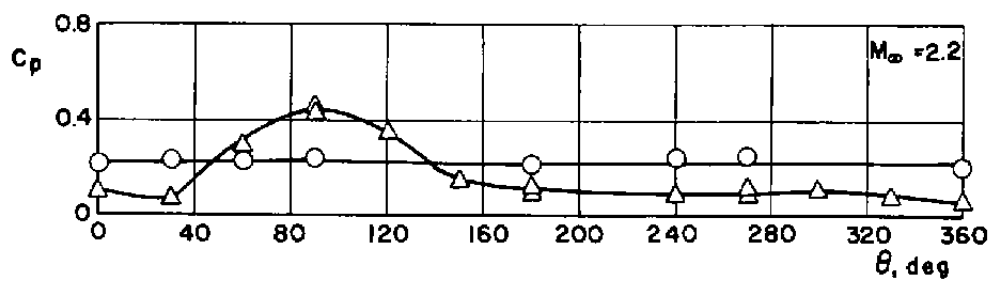
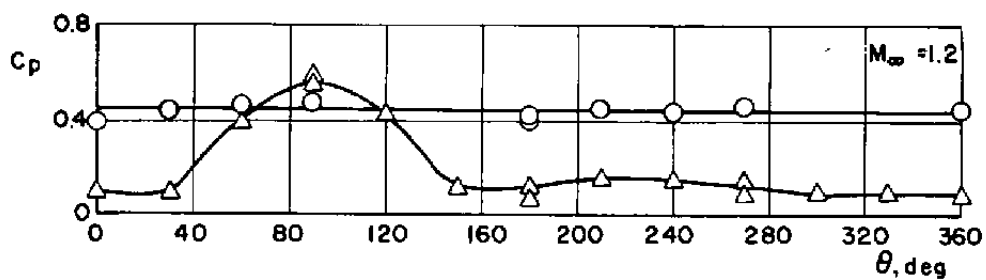
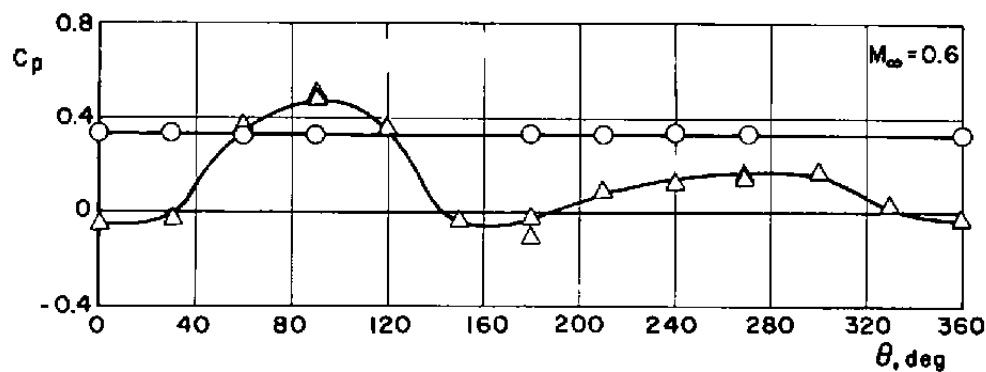
Fig. 12 Circumferential Centerbody Pressure Distribution for Two Model Stations,
 $x/D = 4.182$ and 6.474



b. $x/D = 6.474$, $\alpha = 30$ deg, and $\beta = 0$ deg

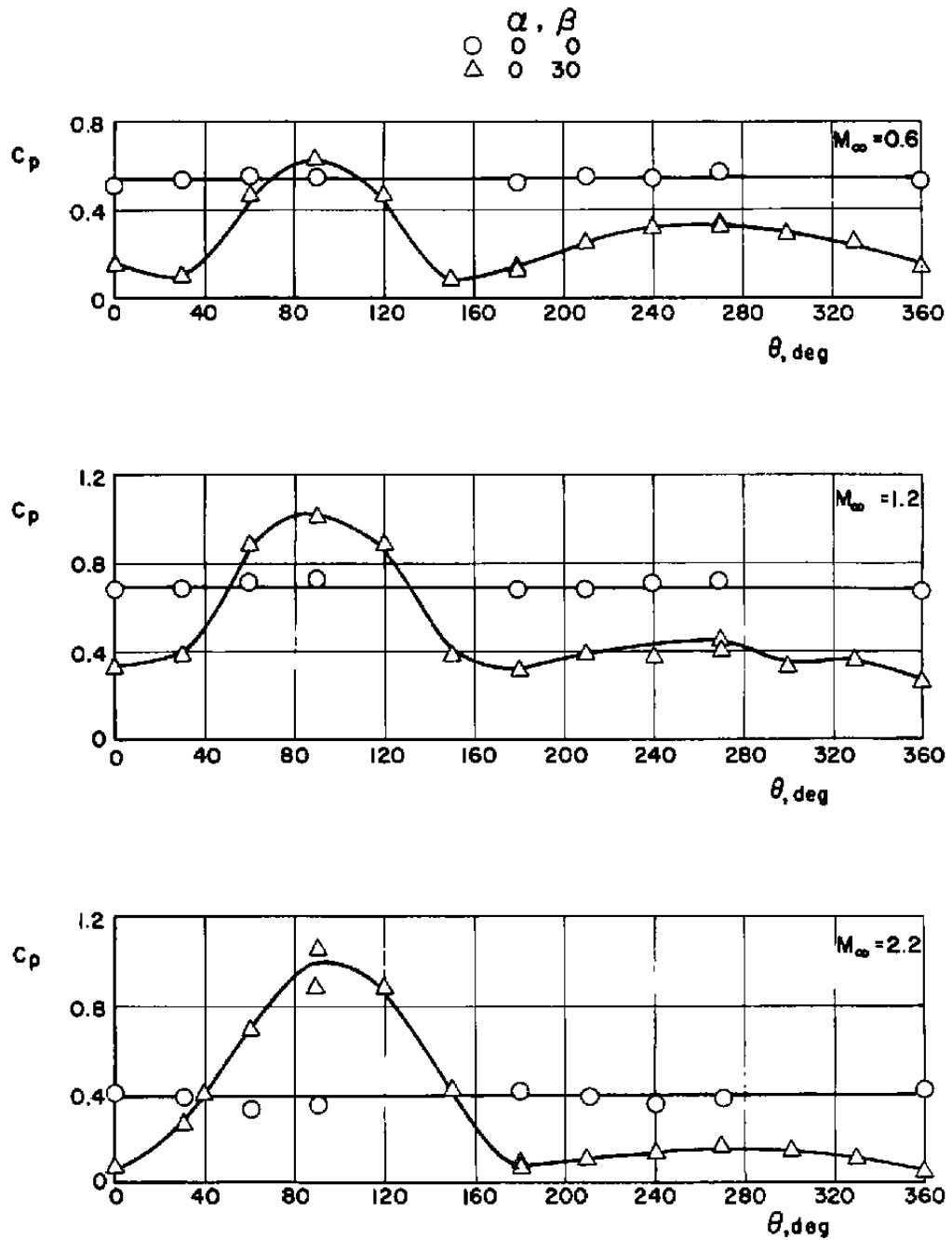
Fig. 12 Continued

	α	β
○	0	0
△	0	30



c. $x/D = 4.182$, $\alpha = 0 \text{ deg}$, and $\beta = 30 \text{ deg}$

Fig. 12 Continued



d. $x/D = 6.474$, $\alpha = 0^\circ$, and $\beta = 30^\circ$

Fig. 12 Concluded

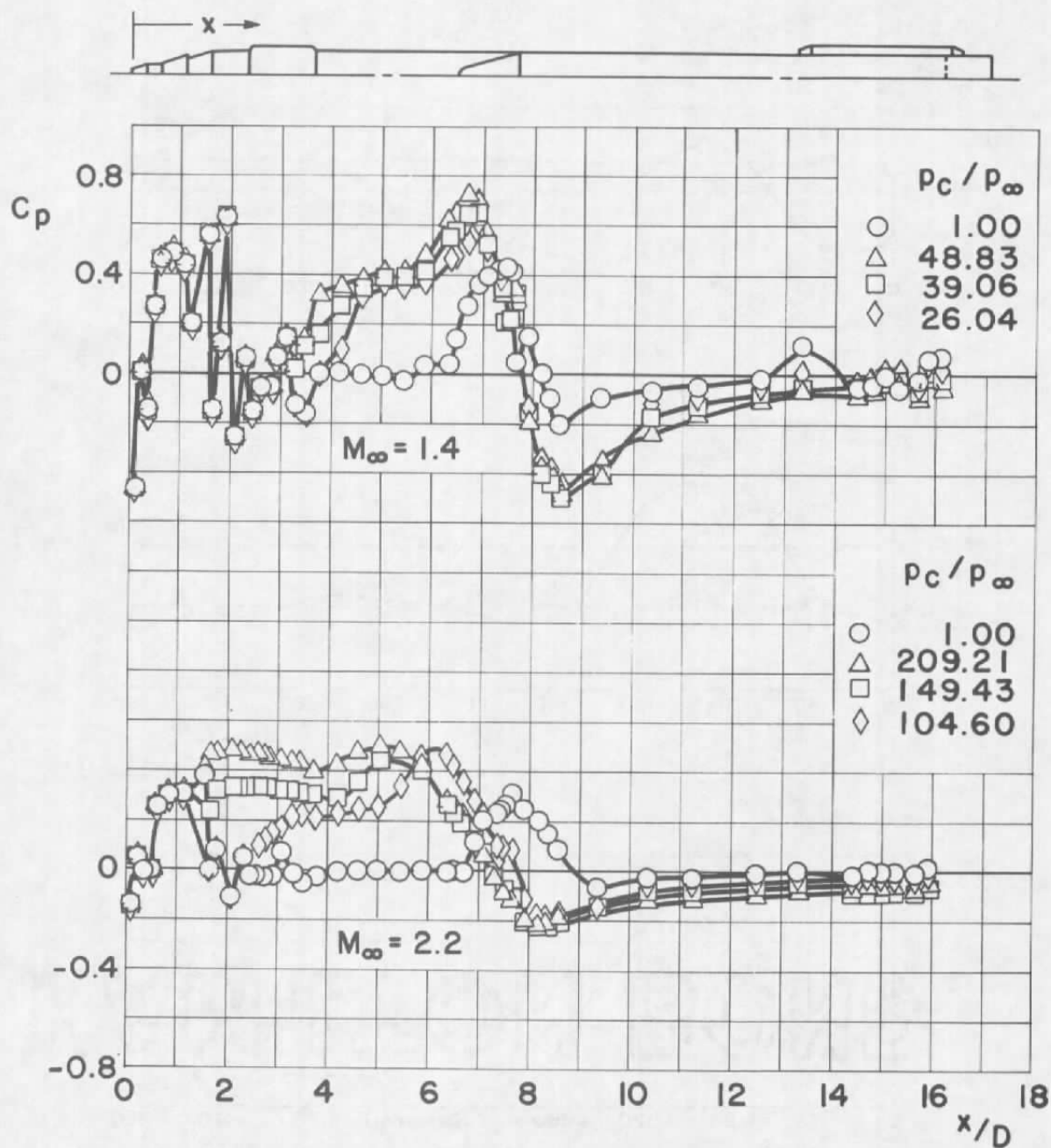
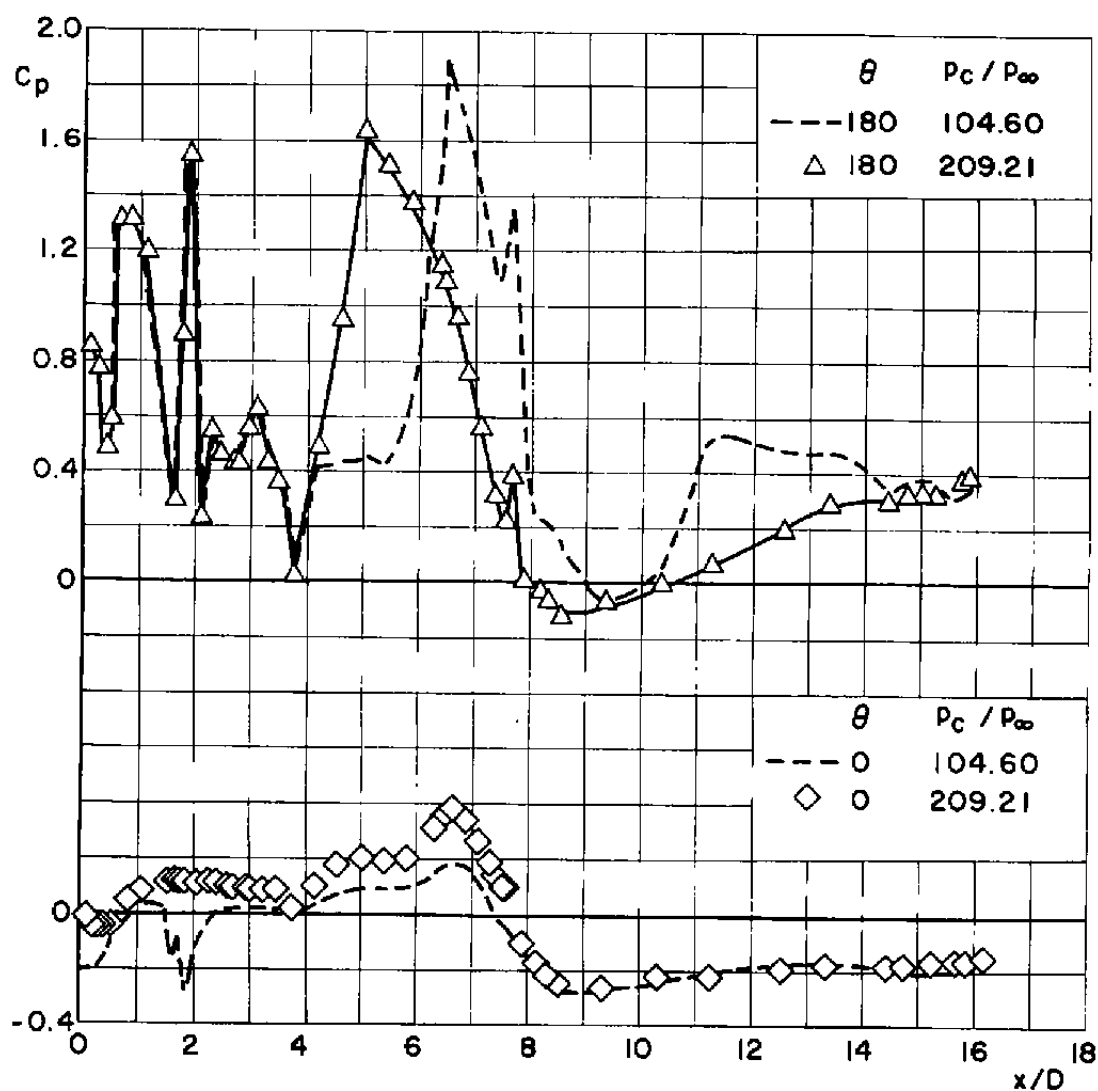
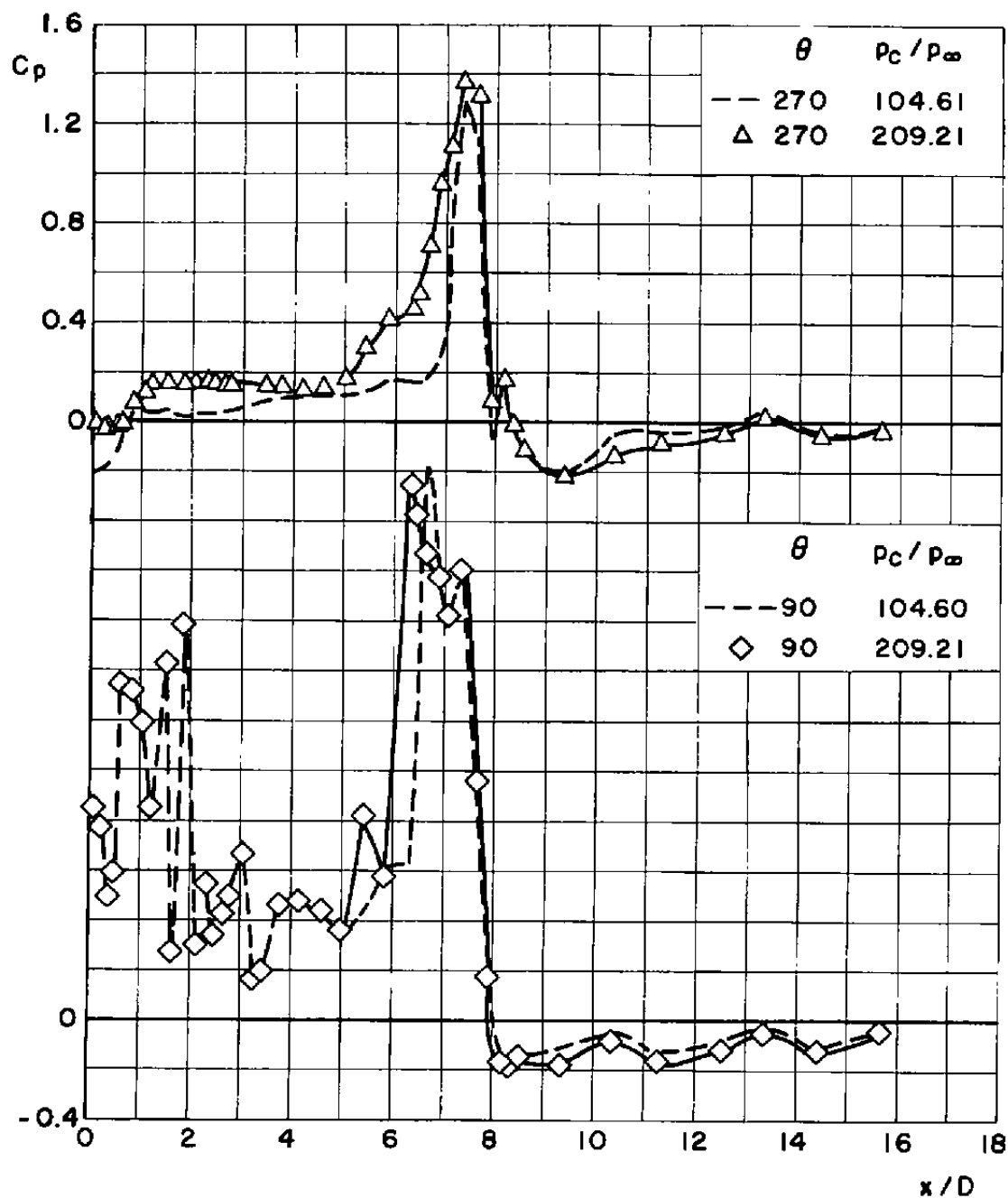


Fig. 13 Centerbody Pressure Distribution for Two Off-Trajectory Pressure Ratios,
 $\theta = 180$ deg, $\alpha = 0$ deg, and $\beta = 0$ deg, Mach Numbers 1.4 and 2.2



a. $\alpha = 30$ deg and $\beta = 0$ deg

Fig. 14 Centerbody Pressure Distribution for Two Off-Trajectory Pressure Ratios at Two Model Attitudes, Mach Number 2.2



b. $\alpha = 0$ deg and $\beta = 30$ deg

Fig. 14 Concluded

UNCLASSIFIED

Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author) Arnold Engineering Development Center, ARO, Inc., Operating Contractor, Arnold Air Force Station, Tennessee		2a. REPORT SECURITY CLASSIFICATION UNCLASSIFIED	
		2b. GROUP N/A	
3. REPORT TITLE WIND TUNNEL INVESTIGATION TO DETERMINE PRESSURE DISTRIBUTION CHARACTERISTICS OF A 0.03-SCALE MODEL OF THE TITAN III/MOL LAUNCH CONFIGURATION DURING THE ABORT SEQUENCE AT MACH NUMBERS FROM 0.60 TO 3.00			
4. DESCRIPTIVE NOTES (Type of report and inclusive dates) March 13 to April 24, 1967			
5. AUTHOR(S) (First name, middle initial, last name) D. A. MacLanahan, Jr. and M. L. Homan, ARO, Inc.			
6. REPORT DATE July 1967		7a. TOTAL NO OF PAGES 37	7b. NO OF REFS 3
8a. CONTRACT OR GRANT NO. AF40(600)-1200		9a. ORIGINATOR'S REPORT NUMBER(S) AEDC-TR-67-141	
b. System 632A and 623A			
c. Program Elements 63409404 and 63409304		9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report) N/A	
d.			
10. DISTRIBUTION STATEMENT This document is subject to special export controls and each transmittal to foreign governments or foreign nationals may be made only with prior approval of SAMSO (SSBD), Los Angeles AF Station, California.			
11. SUPPLEMENTARY NOTES Available in DDC.		12. SPONSORING MILITARY ACTIVITY SAMSO (SSBD), Air Force Systems Command, Los Angeles AF Station, California	
13. ABSTRACT A 0.03-scale model of the Titan III/Manned Orbiting Laboratory (MOL) launch vehicle was tested in Tunnels 16T and 16S of the Propulsion Wind Tunnel Facility at Mach numbers from 0.6 to 3.0 to determine pressure distributions on the airborne vehicle during the abort sequence. Test results show that the jet, simulating thrust termination, forces the solid-propellant motor induced normal shock forward. Disturbances caused by the jet produce an increase in the pressure coefficient in the region ahead of the jet and a decrease in the pressure coefficient in the region aft of the jet. This document is subject to special export controls and each transmittal to foreign governments or foreign nationals may be made only with prior approval of SAMSO (SSBD), Los Angeles AF Station, California.			

KEY WORDS

LINK A

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LINK C

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ROLE

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Titan III/MOL
0.03-scale model
launch vehicles
solid-rocket motors
wind tunnel testing
pressure testing
abort sequence
subsonic flow
supersonic flow